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CRANFIELD

EXTERNAL NOISE LEVEL ON LANCASTER PA474

by

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SUMMARY

Measurements of the external noise level on the Lancaster aircraft PA474 were made at 10,000 feet at two flight speeds for various Merlin engine conditions. In addition the effect on the noise of running one of the Budworth gas turbines, used for the suction wing experiment, was determined. The results show that the maximum sound pressure level found in flight was 124 DB, corresponding to an effective value of $\frac{u'}{U} = 1.2 \times 10^{-3}$, where u' is the sound particle velocity.

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CONTENTS

1. Introduction

As a result of the visit of Dr. Pfenninger and Dr. Moore in June, 1962, regarding the effect of noise on transition, it was decided, at a meeting of the Lancaster progress committee, that the external noise level in the vicinity of the test wing on the aircraft should be obtained.*

2. Instrumentation

The transducer used was a Brüel and Kjaer Type 4133 condenser microphone fitted with a 4 mm bent coupling tube (1 inch long) connected to a 4 mm flush static hole on the elliptic nosed, 1 inch diameter probe (Fig. 2). The microphone was mounted in polyurethane foam to avoid vibration. The coupling tube was fitted with a steel wool plug to damp out acoustic resonances of the tube. The 4 mm tube was the largest diameter normally supplied which gave the best high frequency response.

As the levels to be obtained were to be representative of conditions at the test fin surface, the probe needed to be placed near the fin, while not interfering with the traversing or fly protection gear. On examination, only two locations, namely (a) forward of the tip and (b) forward of the root fence, appeared reasonable. The top position (Fig. 1) was favoured because it was the more distant from the fuselage boundary layer. The probe was mounted parallel to the fuselage datum and was, therefore, in line with the undisturbed flow at the lower test indicated airspeed of 150 knots (equivalent to a true air speed of 292 ft/sec.). The microphone hole was 28 inches forward of the projected leading edge of the test fin.

A test made in a 27 inch by 44 inch open jet tunnel showed that transition occurred at least 15 inches aft of the static hole at 100 ft/sec. Since the tunnel had a high level of turbulence, it seemed reasonable to assume that, under flight conditions, transition would still be well aft of the hole.

The damping of the coupling tube was adjusted, using a standard technique outlined by Brüel and Kjaer which involved inserting the probe into a small calibration chamber housing a dummy microphone and an earphone. By exchanging the position of the microphone in the probe with the dummy microphone, a response curve of the coupling tube was obtained. The size and position of the steel wool plug was adjusted until the flattest response, over the range 40 c/s to 10 Kc/s, was attained.

A Brüel and Kjaer type 2107 frequency analyser was used to amplify the signal from the microphone via a cathode follower (Type 2614).

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* Dr. Pfenninger also discussed the effects of noise created at the suction slit opening. It might, therefore, be desirable to take measurements of this noise on the test wing at some later date. However, a preliminary investigation showed that the noise from the Budworth gas turbine was relatively small and hence the effect of noise from this source was felt to be of little concern.

A vibration resistant mounting, utilizing Equiflex (10 lb rating, medium duty) anti-vibration pads, was used to mount the frequency analyser in the mid-fuselage section of the aircraft.

The noise signal was recorded on an instrumentation tape recorder (Flexonics A4011) using the Direct Recording Process. This gave an overall frequency response of $\pm$ 3 DB for 50 to 12,000 cycles/sec. The measured response is shown in Fig. 3. The airborne tapes were cut into 18 foot loops and were replayed on a Flexonics Type 409 ground replay machine.

After the flight tests, a free field calibration of the complete microphone probe was made against a standard half-inch microphone fitted with the standard protection grid. A tape loop from one of the flight recordings was used to excite a six watt speaker located three feet from the microphone. Since a $\frac{1}{3}$ octave band analyser was used and no marked discrete frequencies were present, no trouble from standing waves was experienced.

3. Flight Tests

An earlier test to obtain noise levels in flight was considered at the time to be unsuccessful since, on playback, the signal appeared to be similar to that obtained from a turbulent boundary layer. It was noticed that the mounting of the probe contained a step downstream of the static hole. Since this step was not required for any structural purpose, it was decided to remake the mounting tube to a preferred standard of finish.

The doubts expressed on the earlier test led to the use of a transition ring, which could be removed in flight, thereby allowing any consequent change in signal to be observed. A thread, in the form of a ring, was attached forward of the microphone hole and was broken manually, at the appropriate time, by a string led to the forward hatch of the aircraft.

The noise tests were made as detailed in Fig. 4. Each test was recorded for 25 seconds after conditions were held constant for 1 to 2 minutes.

When the transition ring was removed on the first flight at 150 knots I.A.S. with the Merlin engines throttled back, a distinct change in signal level was noted, both audibly and from an oscilloscope. It was, consequently, assumed that the flow over the probe was laminar and that the transducer recorded true noise.

4. Ground Tests

Two ground tests were made with two Merlin engines set at cruise conditions. Two recordings were taken, one with the microphone rigidly mounted as for flight and the other with it hand held in approximately the same position. The runs were made to determine if vibration transmitted by the structure was likely to give an additional spurious signal.

One test was made with a Budworth turbine only set at maximum R.P.M. It was noted that the sound level was considerably lower than that of the Merlin engines.

One further test was made with two of the Merlin engines throttled back. The standard microphone was located near the wing tip and the results of this test were used in an attempt to distinguish between sources of aerodynamic noise and engine noise in the comparable flight case.

5. Analysis

The signal from the tape loops was analysed in $\frac{1}{3}$ octave bands using a Brüel and Kjaer Type 2107 frequency analyser. The readings were analysed over a frequency range of 40 c/s to 10 Kc/s.

From the free field calibration of the probe (Fig. 5) and the known response of the tape equipment (Fig. 3), the fourteen tests were plotted in groups (Fig. 6 - 9) as sound pressure levels per cycle of bandwidth versus frequency.

6. Discussion

There was negligible transmission of vibration through the structure and the microphone mounting, as shown by the two comparative ground tests (Fig. 6, Runs 1 and 2). The ground tests were not, of course, directly comparable with the flight tests since they were made on a hangar apron and only two Merlin engines were run.

After the flights were concluded, a wind tunnel test was performed to check the incidence tolerance of the probe for laminar flow over the microphone static hole. The output of the microphone was measured and an attempt was made to determine the transition incidence but the large signal produced by the tunnel noise prevented this. Flow visualisation, using the naphthaline technique, clearly indicated that the flow in the vicinity of the hole was laminar between -1° and $+2^{\circ}$ incidence. These tests did not show at what stage the flow became turbulent because transition was not clearly defined. The tunnel was run at a nominal speed of 200 feet/sec.

In the flight tests the probe was at 0° and -2° incidence to the undisturbed flow at 150 and 200 knots I.A.S. (292 ft/sec and 385 ft/sec. T.A.S.) respectively. If there are small variations in the local flow direction of up to $\pm 1^{\circ}$, laminar flow should exist over the probe at the lower test speed but the state of the boundary layer at the higher speed was not known. The flight tests, with and without a transition ring, at 150 knots I.A.S., with the engines throttled back (Fig. 7) also indicated that the flow over the probe was laminar for that particular condition. However, at higher flight speeds or noisier engine conditions, it is not clear that the flow remained laminar over the microphone hole. Under these conditions of flight, the spectral density is dominated by low frequency noise of propellers and engines, and the added contribution to the overall noise level of the high frequency pressure fluctuations on the turbulent boundary layer flow over the probe would be small.

In Fig. 8, the peak at 60 c/s corresponds to the three bladed propeller frequency with the engine R.P.M. of 2850 and a gearbox ratio of 0.42 : 1. Early work (Ref. 1) indicates that the exhaust stubs of an in-line piston engine contribute two predominant peaks, the lower resulting from the number of explosions per revolution from one side of the engine and the higher having twice this value. The peaks at 140 c/s and

285 c/s correspond to the exhaust frequency of 3 and 6 explosions per revolution. Similar peaks may be distinguished at other R.P.M. conditions.

The contribution of the noise from the Budworth gas turbine in the ground test (Fig. 6) was negligible so it is not surprising that the flight tests, with and without the Budworth turbine running (Fig. 8), were virtually identical.

The three pairs of curves in Figs. 8 and 9, two taken at the same flight speed and two at the same engine R.P.M., show the shift in the low frequency part of the spectrum (below 600 c/s) with engine R.P.M. and the shift in the higher frequency part of the spectrum from 600 c/s to 4 Kc/s) with flight speed. In particular, the peak near 900 c/s at 150 knots I.A.S., changing to 1200 c/s at 200 knots I.A.S., appears to be associated with vortex shedding from the $\frac{3}{4}$ inch diameter mounting strut 20 inches behind the microphone.

Fig. 4 shows the measured pressure levels in microbars and decibels relative to a sound pressure level of 0.0002 dyne/cm^2 and the effective sound particle speed to flight speed ratio, based on the assumption of plane waves. The highest ratio obtained in flight was on the high R.P.M. test at 150 knots I.A.S. which gave a reading of 328 μ bar. This yields 124 DB and $\frac{u'}{U} = \frac{0.36}{292} = 0.12\%$, which, according to Pfenninger, has a similar effect to a turbulence level of the same magnitude.

7. Conclusions

In Dr. Pfenninger's experiments (Ref. 2), in which the front half of the wing had the suction slots closed and suction was applied to the rear half of the wing at a wing chord Reynolds number of 7×10^6 and less, the boundary layer over the front half of the wing became unstable in the frequency range for Tollmien-Schlichting waves at a critical $\frac{u'}{U}$ of 0.0002. This was the only case, according to Pfenninger, in which transition due to critical sound pressures could be correlated to Tollmien-Schlichting stability theory but it is important because of the very low value of $\frac{u'}{U}$ that caused instability.

For a wing chord Reynolds number of 8×10^6, with static pressure holes and unsucked slots sealed and with minimum suction applied, Pfenninger found that the critical $\frac{u'}{U}$ = 0.0012 to 0.0018, over a wide range of frequencies. In establishing this criterion, Pfenninger used a frequency range from 400 c/s to 1000 c/s.

When comparing the measured values of $\frac{u'}{U}$ on the Lancaster aircraft with Pfenninger's criteria, it must be remembered that, although the size and chord R_e of the test wings are similar, Pfenninger's results were obtained in a wind tunnel test and the corresponding criteria may be different in flight. The measured value of $\frac{u'}{U}$ of 0.0006, corresponding to the cruise condition at 150 knots I.A.S. on the Lancaster, is greater than the critical value associated with noise at the Tollmien-Schlichting amplification frequency established by Pfenninger, although this only applied to a partially sucked wing. However, this value of $\frac{u'}{U}$ of 0.0006 is less than the critical value for the fully sucked wing with minimum suction applied.

It is clear from the above criteria and the noise present on the Lancaster aircraft, that all static holes and unsucked slots must be carefully sealed for tests involving minimum suction. It must also be noted that the measured values of $\frac{u'}{U}$ in the Lancaster tests were obtained at one location only and it cannot be assumed that these values will be constant over the whole test wing.

It would appear that by operating the Lancaster aircraft under different flight conditions, a sufficient increase in noise beyond the above critical values of $\frac{u'}{U}$ could be obtained, so that the increase in suction to obtain full chord laminar flow could be established. At 200 knots I.A.S. cruise, the maximum value of $\frac{u'}{U}$ measured was 0.0010, which is also just below the critical value for minimum suction.

8. Acknowledgements

The author wishes to thank Professor G. M. Lilley and Mr. M. Gaster of the Department of Aerodynamics for their advice and assistance in the programme. Mr. M. A. Perry and Mr. D. Hyde of the Department of Flight were responsible for the recording equipment and wind tunnel tests respectively.

9. References

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2. Pfenninger, W. and Bacon, J.W. Influence of acoustical disturbances on the behaviour of a swept laminar suction wing. Laminar Flow Control Presentation, Northrop Corporation, Hawthorne, Calif., NB 62-105, May, 1962. Revised August, 1962.

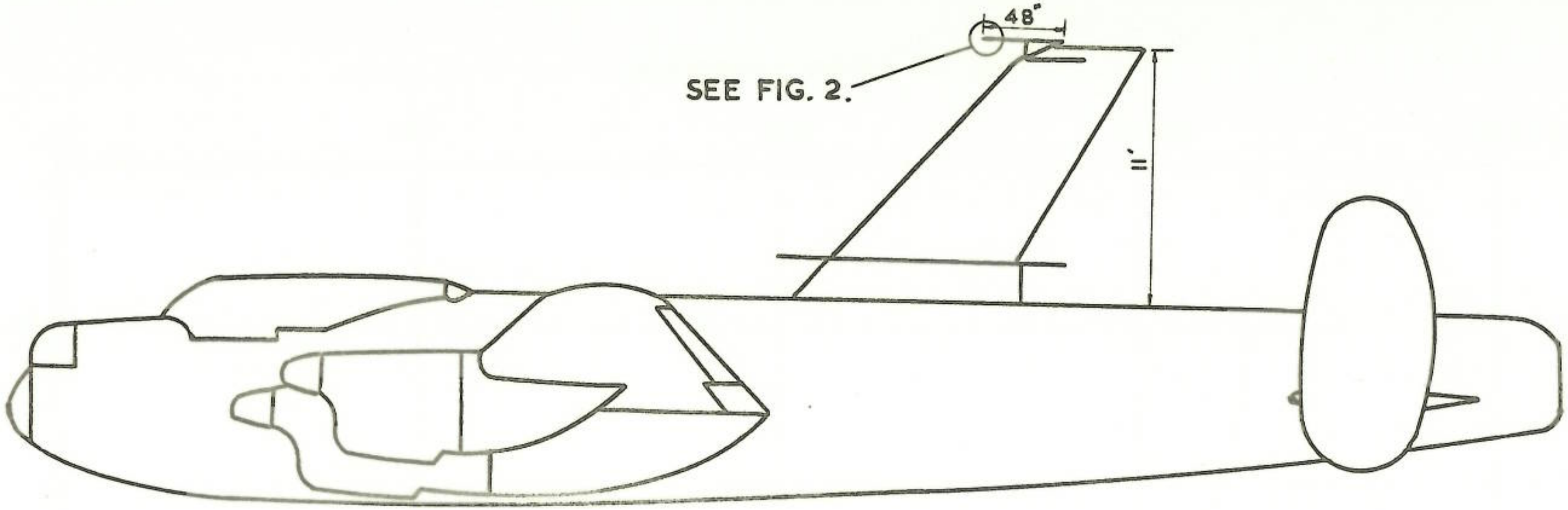


FIG.1 LOCATION OF PROBE

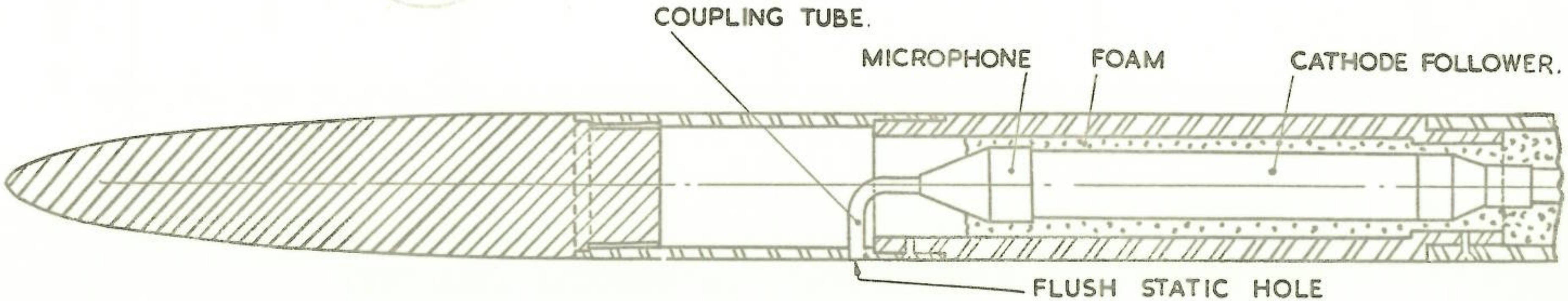
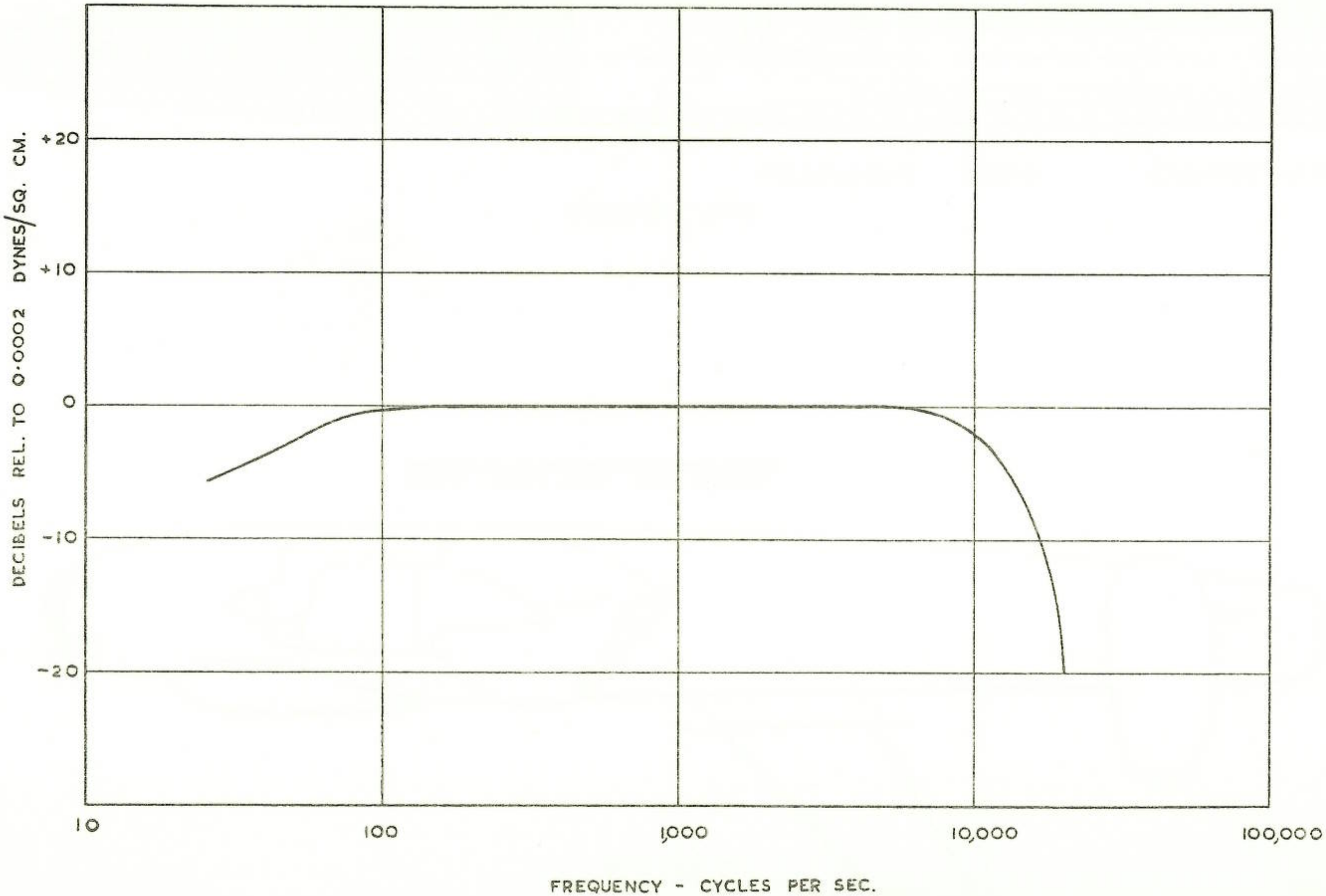


FIG.2 MOUNTING OF MICROPHONE

FIG. 3.

FREQUENCY RESPONSE OF DIRECT RECORD CHANNEL AT 7½"/SEC.



Run	Indicated Airspeed Knots	Merlin Engines			Budworth Gas Turbine R.P.M.	Comments	R.M.S. Pressure per cycle of band-width in μ bar	R.M.S. Pressure per cycle of band-width in DB rel. 0.0002 dynes/cm^2	Sound particle velocity/free stream velocity $\frac{u'}{U}$
		Condition	R.P.M.	Boost Pressure P.S.I.					
1	-	Two port only	2600	0	-	Microphone rigidly mounted	384	126	-
2	-	Two port only	2600	0	-	Microphone hand held	333	124	-
3	-	Two port only	2000	3	-		160	118	-
4	-	-	-	-	35,000		6.6	90	-
5	150	Throttled back	1750	0	-	Transition ring on	164	118	0.0006
6	150	Throttled back	1750	0	-		79	112	0.0003
7	150	Throttled back	1750	0	-		77	112	0.0003
8	150	Throttled back	2100 1850 1900 1900	0	35,000		73	111	0.0003
9	150	High R.P.M.	2850	9	-		328	124	0.0012
10	150	Unsynchronised	2850 2600 2800 2650	9	-	Each engine at different R.P.M.	286	123	0.0011
11	150	Cruise	2200	6	-		169	119	0.0006
12	150	Cruise	2200	6	35,000		167	118	0.0006
13	200	Cruise	2850	9	-		337	124	0.0010
14	200	Cruise	2850	9	35,000		304	124	0.0009

FIG. 4. TABLE OF FLIGHT AND GROUND TESTS

FIG 5

FREE FIELD CALIBRATION OF PROBE

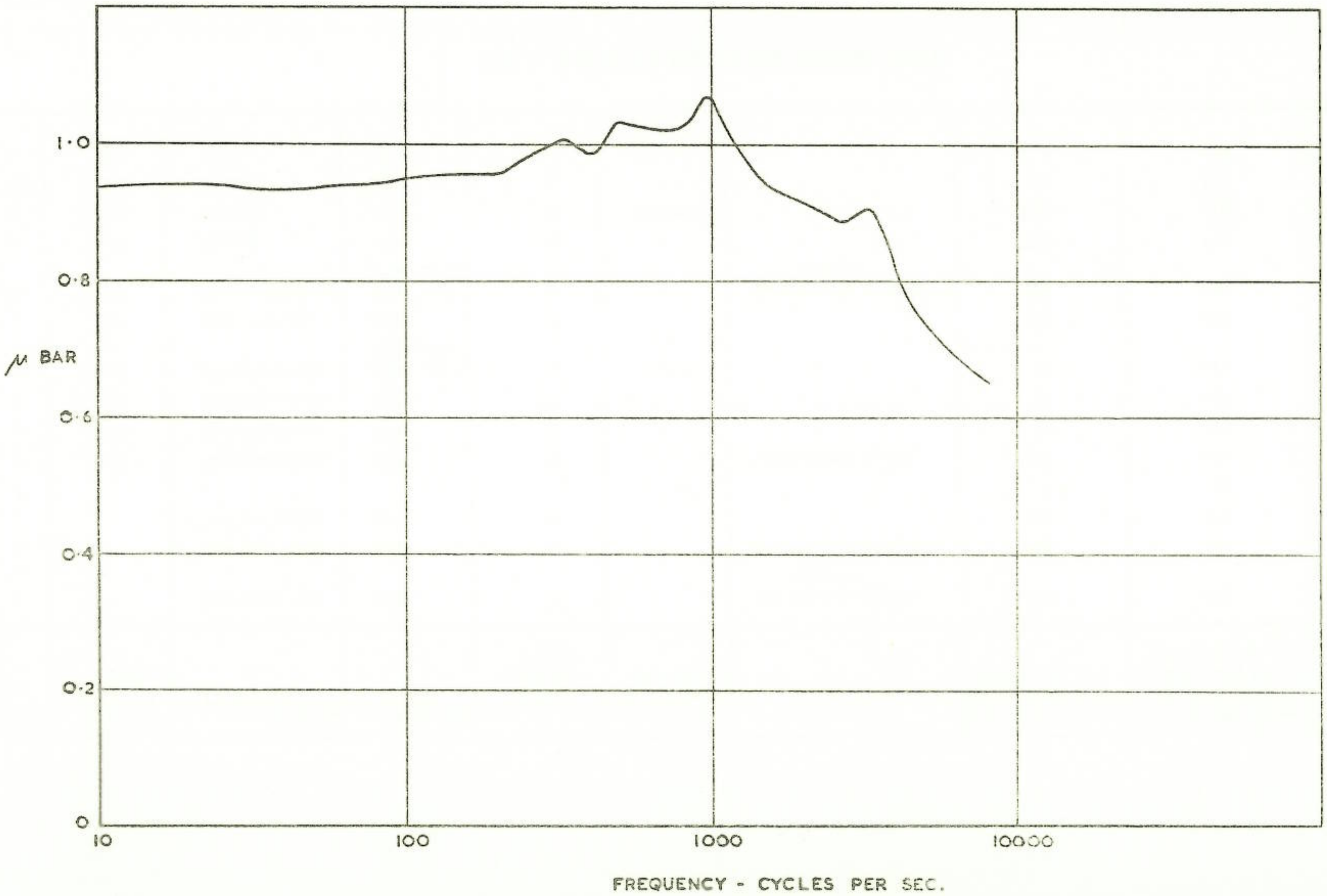


FIG. 6.

NOISE SPECTRA.

GROUND RUNS

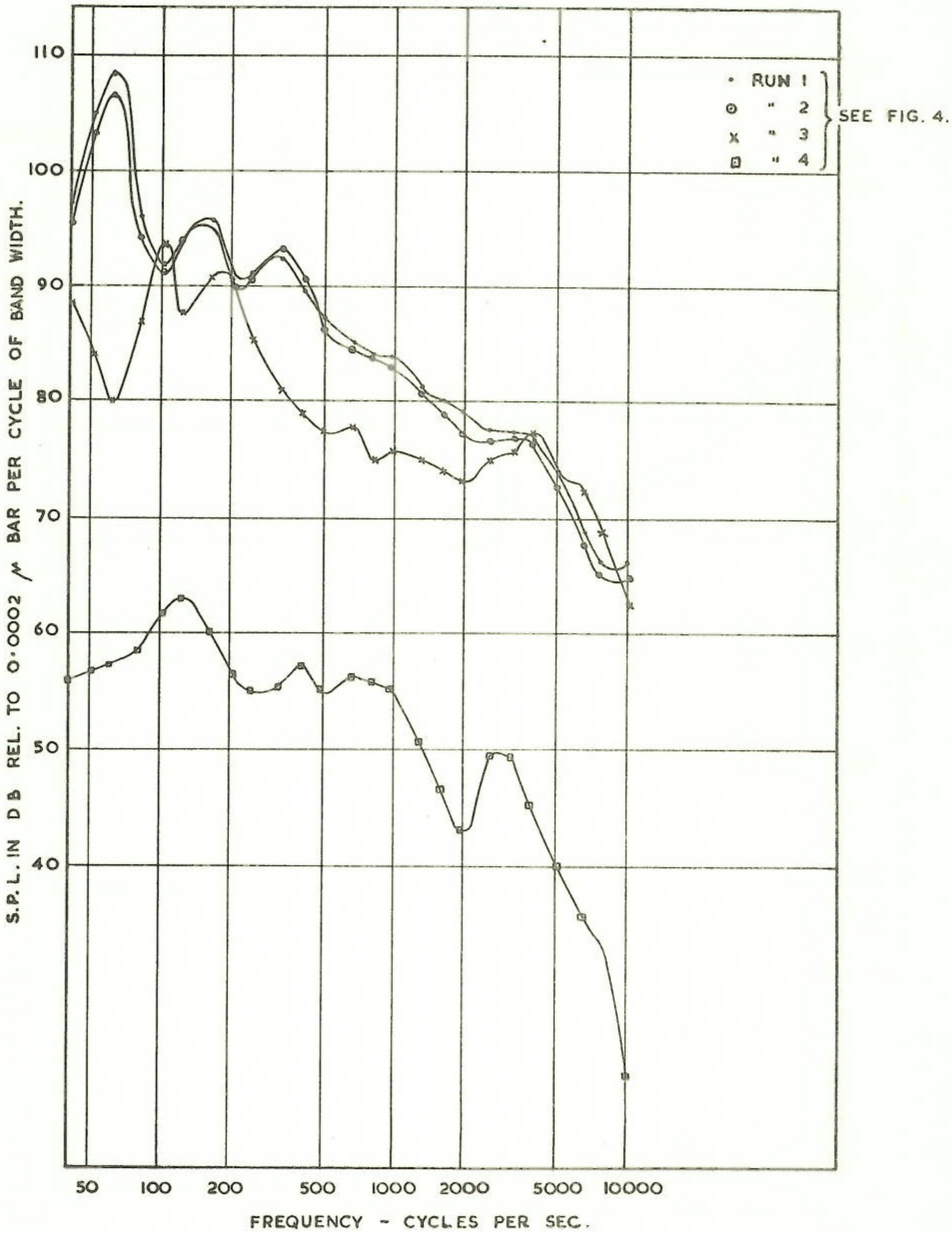


FIG. 7.

NOISE SPECTRA.

150 KNOTS THROTTLED BACK 1750 R.P.M.

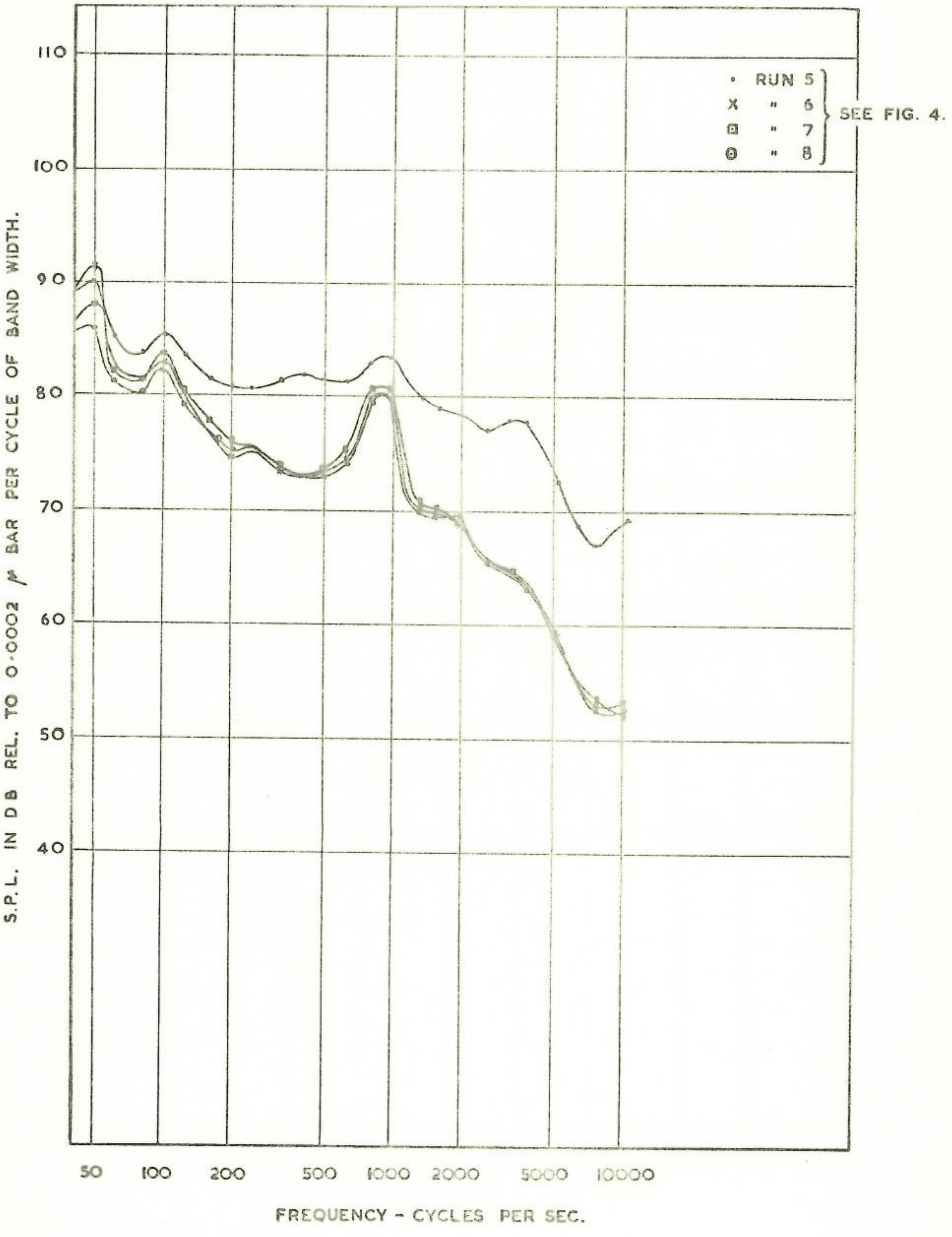


FIG. 8.

NOISE SPECTRA.

150 KNOTS MERLINS CRUISE (2200 R.P.M. & 2850 R.P.M.)

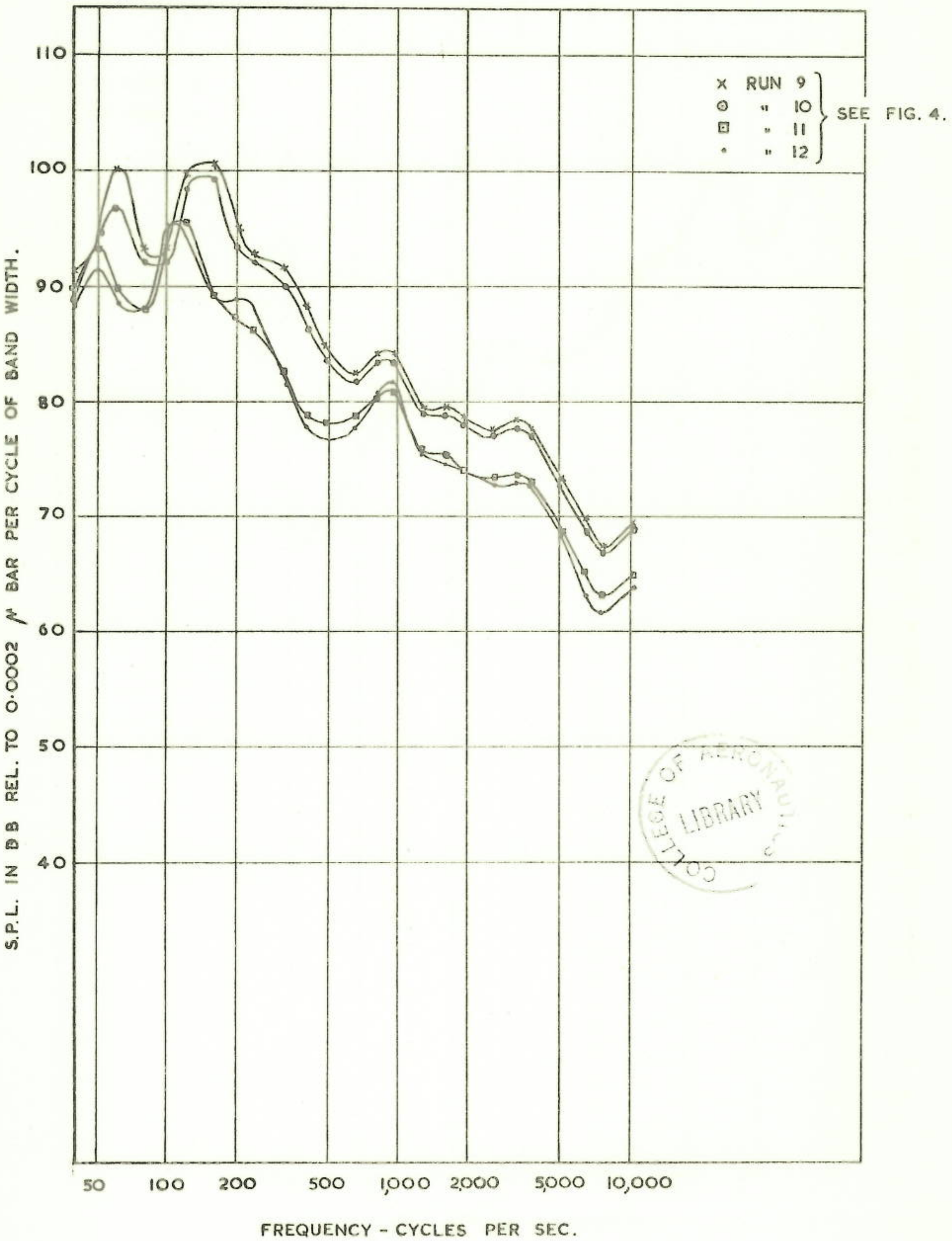


FIG. 9.

NOISE SPECTRA

200 KNOTS (MERLINS 2850 R.P.M.)